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RESEARCH MEMORANDUM

LIFT, DRAG, AND PITCHING MOMENT OF LOW-ASPECT-RATIO WINGS AT SUBSONIC AND SUPERSONIC SPEEDS - PLANE TAPERED WING OF ASPECT RATIO 3.1 WITH 3-PERCENT-THICK ROUNDED-NOSE SECTION

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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RESEARCH MEMORANDUM

LIFT, DRAG, AND PITCHING MOMENT OF LOW-ASPECT-RATIO WINGS AT SUBSONIC AND SUPERSONIC SPEEDS - PLANE TAPERED WING OF ASPECT RATIO 3.1 WITH 3-PERCENT-THICK ROUNDED-NOSE SECTION

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SUMMARY

A wing-body combination having a plane tapered wing of aspect ratio 3.1, taper ratio of 0.39, and 3-percent-thick, rounded-nose sections in streamwise planes has been investigated at both subsonic and supersonic Mach numbers. The lift, drag, and pitching moment of the model are presented for Mach numbers from 0.60 to 0.92 and from 1.20 to 1.90 at Reynolds numbers of 1.5 million, 2.4 million, and 3.8 million. (At a Reynolds number of 3.8 million the maximum test Mach number was limited to 1.70 because of wind-tunnel power limitations.)

INTRODUCTION

A research program is in progress at the Ames Aeronautical Laboratory to ascertain experimentally at subsonic and supersonic Mach numbers the characteristics of wings of interest in the design of high-speed fighter airplanes. The effects of variation in plan form, twist, camber, and thickness are being investigated. The results of this program to date are presented in references 1 through 14.

This report is one of a series pertaining to this program and presents results of tests of a wing-body combination having a plane tapered wing of aspect ratio 3.1 and taper ratio of 0.39. The model is the same as that reported in reference 7, except that the 3-percent-thick, biconvex section of reference 7 was modified. This modification consisted of replacing the portion of the biconvex section, forward of the midchord location, with an elliptical profile. The tangent to the airfoil section at the 50-percent-chord position was parallel to the chord

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line. Figure 1 shows pictorially the extent of this modification. As in references 1 through 14, the data herein are presented without analysis to expedite publication.

NOTATION

b	wing span
$\bar{c}$	mean aerodynamic chord $\left(\dfrac{\int_0^{b/2} c^2\,dy}{\int_0^{b/2} c\,dy}\right)$
c	local wing chord
l	length of body, including portion removed to accommodate sting
$\frac{L}{D}$	lift-drag ratio
$\left(\frac{L}{D}\right)_{max}$	maximum lift-drag ratio
M	Mach number
q	free-stream dynamic pressure
R	Reynolds number based on mean aerodynamic chord
r	radius of body
r_o	maximum body radius
S	total wing area, including area enclosed by body
x	longitudinal distance from nose of body
y	distance perpendicular to vertical plane of symmetry
α	angle of attack of the body axis, degrees
C_D	drag coefficient $\left(\frac{\text{drag}}{qS}\right)$
C_L	lift coefficient $\left(\frac{\text{lift}}{qS}\right)$
C_m	pitching-moment coefficient referred to quarter point of mean aerodynamic chord $\left(\frac{\text{pitching moment}}{qS\bar{c}}\right)$

$\frac{dC_L}{d\alpha}$ slope of the lift curve measured at zero lift, per degree

$\frac{dC_m}{dC_L}$ slope of the pitching-moment curve measured at zero lift

APPARATUS

Wind Tunnel and Equipment

The experimental investigation was conducted in the Ames 6- by 6-foot supersonic wind tunnel. In this wind tunnel, the Mach number can be varied continuously and the stagnation pressure regulated to maintain a given test Reynolds number. The air is dried to prevent formation of condensation shocks. Further information on this wind tunnel is presented in reference 15.

For the present investigation a sting bent 10° in the direction of positive lift was used to mount the model in the wind tunnel, the diameter of the sting being about 93 percent of the diameter of the body base. Due to the bend in the sting, the model center line was displaced laterally from the tunnel center line about 4 inches. The pitch plane of the model support was horizontal. The 4-inch diameter, four-component strain-gage balance, described in reference 16, was enclosed within the body of the model and was used to measure the aerodynamic forces and moments.

Model

A front and plan view of the model and certain model dimensions are given in figure 2. Other important geometric characteristics of the model are as follows:

Wing

Aspect ratio	3.1
Taper ratio	0.39
Airfoil section (streamwise)	3-percent-thick, modified biconvex (fig. 1)
Total area, S, square feet	2.425
Mean aerodynamic chord, $\bar{c}$, feet	0.944
Dihedral, degrees	0
Camber	None

[redacted]

Twist, degrees 0
Incidence, degrees 0
Distance, wing-chord plane to body axis, feet 0

Body

Fineness ratio (based upon length, l; fig. 2) 12.5
Cross-section shape . . . . Circular
Maximum cross-sectional area, square feet 0.1235
Ratio of maximum cross-sectional area to wing area 0.0509

The wing contour of the present model was obtained by covering the solid steel wing of reference 7 with a tin-bismuth alloy. The body spar was steel and was covered with aluminum to form the body contour. The surfaces of the wing and body were polished smooth.

TESTS AND PROCEDURE

Range of Test Variables

The characteristics of the model (as a function of angle of attack) were investigated for a range of Mach numbers from 0.60 to 0.92 and from 1.20 to 1.90. The data were obtained at Reynolds numbers of 1.5 million, 2.4 million, and 3.8 million. (Tests at a Reynolds number of 3.8 million were limited to a maximum test Mach number of 1.70 because of wind-tunnel power limitations.)

Reduction of Data

The test data have been reduced to standard NACA coefficient form. Factors which could affect the accuracy of these results, together with the corrections applied, are discussed in the following paragraphs.

Tunnel-wall interference.- Corrections to the subsonic results for the induced effects of the tunnel walls resulting from lift on the model were made according to the method of reference 17. The numerical values of these corrections (which were added to the uncorrected data) were obtained from

$$\Delta\alpha = 0.57\, C_L$$

$$\Delta C_D = 0.0100\, C_L^2$$

No corrections were made to the pitching-moment coefficients.

The effects of constriction of the flow at subsonic speeds by the tunnel walls were taken into account by the method of reference 18. This correction was calculated for conditions at zero angle of attack and was applied throughout the angle-of-attack range. At a Mach number of 0.90, this correction amounted to a 2 percent increase in the Mach number and in the dynamic pressure over that determined from a calibration of the wind tunnel without a model in place.

For the tests at supersonic speeds, the reflected Mach wave, which originated at the nose of the body, did not cross the model. No corrections were required, therefore, for tunnel-wall effects.

Stream variations.- Tests at subsonic speeds of the present model in both the normal and inverted positions have indicated a stream inclination of -0.10° and a stream curvature capable of producing a pitching-moment coefficient of -0.002 at zero lift. The data of the present report have been corrected for the effects of these stream irregularities. No measurements have been made of the stream curvature in the yaw plane. At subsonic speeds, the longitudinal variation of static pressure in the region of the model is not known accurately at present, but a preliminary survey has indicated that it is less than 2 percent of the dynamic pressure; consequently, no correction for this effect was made.

Tests of the present model at supersonic speeds in both the normal and the inverted positions have indicated that an apparent stream inclination of about -0.20° exists at a Mach number of 1.2. This apparent downflow is not believed to be an irregularity in the tunnel free stream, but is believed to be related to the presence of the bent sting used to mount the model in the tunnel. This belief is substantiated by the results of tests of models in both the normal and inverted positions which indicated that no stream inclination exists at a Mach number of 1.2 when the models were mounted on a straight sting. The data of the present report obtained at a Mach number of 1.2 have been corrected for the effect of this apparent stream inclination. A survey of the air stream at supersonic speeds (reference 15) has shown a stream curvature in the yaw plane of the model. The effects of this curvature on the measured characteristics of the present model are not known but are believed to be small as judged by the results of reference 19. The survey of reference 15 also indicated that there is a static-pressure variation in the test section of sufficient magnitude to affect the drag results. A correction was added to the measured drag coefficient, therefore, to account for the longitudinal buoyancy caused by this static-pressure variation. This correction varied from as much as -0.0007 at a Mach number of 1.30 to 0.0006 at a Mach number of 1.70. No buoyancy correction was made at a Mach number of 1.90.[1]

Support interference.- At subsonic speeds, the effects of support interference on the aerodynamic characteristics of the model are not known. For the present tailless model, it is believed that such effects consisted primarily of a change in the pressure at the base of the model. In an effort to correct at least partially for this support interference, the base pressure was measured and the drag data were adjusted to correspond to a base pressure equal to the static pressure of the free stream.

At supersonic speeds the effects of support interference of a body-sting configuration similar to that of the present model are shown by reference 20 to be confined to a change in base pressure. The previously mentioned adjustment of the drag for base pressure, therefore, was applied at supersonic speeds.

It should be noted that the drag coefficients presented are in essence foredrag coefficients, since the drag data do not include the base drag to which a free-flight model would be subject.

RESULTS

The results are presented in this report without analysis in order to expedite publication. The variation of lift coefficient with angle of attack and the variations of pitching-moment coefficient, drag coefficient, and lift-drag ratio with lift coefficient at Mach numbers from 0.60 to 1.90 at the test Reynolds numbers are shown in figure 3. The data presented in figure 3 are tabulated in tables I, II, and III. The results presented in figure 3 for a Reynolds number of 2.4 million have been summarized in figure 4 to show some important parameters as functions of Mach number. Included in figure 4(c) are the values of the maximum lift-drag ratios at subsonic speeds for a Reynolds number of 3.8 million. Also presented in figure 4, for comparison purposes, are corresponding data of reference 7.[2] The slope parameters in this figure have been measured at zero lift.

[1]Results of a static-pressure survey performed after the reduction of the data of the present report and after the publication of reference 7 indicate that at a Mach number of 1.90 a correction of 0.0006 should be added to the drag coefficients presented in the present report and in reference 7 to account for the longitudinal buoyancy.

[2]The pitching-moment data of reference 7 have been referred to the 25-percent mean aerodynamic chord position for presentation in figure 4.

It should be noted that the results presented for the rounded-nose wing (see tabulated data) show significant variation of minimum drag and drag due to lift with Reynolds number at subsonic speeds. These variations are reflected in the lift-drag ratios presented in figures 3(d) and 4(c).

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TABLE I.- AERODYNAMIC CHARACTERISTICS OF THE MODEL
AT A REYNOLDS NUMBER OF 1.5 MILLION

M	α	C_L	C_D	C_m	M	α	C_L	C_D	C_m
0.60	-0.38	-0.030	0.0046	-0.005	0.75	-1.21	-0.088	0.0072	-0.006
	-.65	-.044	.0047	-.003		-2.31	-.165	.0111	-.010
	-.93	-.065	.0054	-.004		-3.40	-.247	.0173	-.017
	-1.19	-.082	.0066	-.005		.15	.008	.0047	0
	-2.27	-.151	.0100	-.010		.44	.029	.0050	-.001
	-3.36	-.224	.0153	-.015		.71	.048	.0053	.001
	-4.43	-.298	.0233	-.021		.99	.069	.0059	.002
	.15	0	.0044	-.002		2.09	.148	.0103	.007
	.43	.025	.0045	-.001		3.19	.234	.0159	.011
	.70	.042	.0051	0		4.27	.313	.0246	.018
	.97	.058	.0060	.001		5.37	.397	.0373	.024
	2.05	.135	.0092	.004		6.46	.482	.0546	.025
	3.14	.207	.0134	.009		8.61	.630	.0973	-.001
	4.22	.285	.0221	.014		10.68	.701	.1411	-.047
	5.29	.358	.0332	.018		12.73	.750	.1830	-.075
	6.37	.433	.0482	.019					
	8.53	.589	.0885	.002	0.80	-.39	-.031	.0052	-.002
	10.63	.698	.1350	-.033		-.94	-.073	.0068	-.005
	12.65	.724	.1721	-.069		-1.22	-.092	.0078	-.006
	14.66	.738	.2067	-.083		-2.33	-.177	.0121	-.010
	18.70	.773	.2893	-.090		-3.42	-.263	.0181	-.018
						.15	.003	.0050	-.001
0.70	-.38	-.029	.0048	-.002		.44	.030	.0050	0
	-.66	-.049	.0050	-.003		.71	.049	.0053	.002
	-.93	-.067	.0060	-.004		.99	.070	.0060	.003
	-1.20	-.084	.0066	-.006		2.10	.156	.0104	.008
	-2.29	-.159	.0107	-.010		3.20	.245	.0164	.013
	-3.38	-.238	.0162	-.016		4.30	.333	.0262	.021
	.15	.003	.0043	-.001		5.41	.426	.0408	.028
	.43	.026	.0045	0		6.51	.515	.0584	.032
	.70	.047	.0047	.001					
	.98	.065	.0056	.002	0.90	-.39	-.032	.0056	-.003
	2.08	.146	.0098	.005		-.67	-.053	.0056	-.006
	3.17	.223	.0148	.010		-.95	-.076	.0063	-.009
	4.25	.300	.0231	.016		-1.23	-.096	.0068	-.012
	5.33	.379	.0352	.020		-2.37	-.199	.0124	-.017
	6.43	.461	.0515	.018		-3.50	-.316	.0225	-.015
	8.57	.603	.0913	.001		.15	.004	.0058	.001
	10.67	.702	.1387	-.042		.44	.029	.0054	.004
	12.69	.732	.1768	-.072		.72	.050	.0056	.006
	14.70	.738	.2094	-.082		1.00	.071	.0060	.009
	16.71	.746	.2419	-.086		2.24	.174	.0107	.016
						3.27	.295	.0195	.014
0.75	-.38	-.030	.0049	-.002		4.40	.415	.0346	.003
	-.66	-.050	.0055	-.004		5.58	.572	.0596	-.035
	-.94	-.070	.0062	-.005		6.66	.648	.0857	-.064

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TABLE I.- CONCLUDED

M	α	C_L	C_D	C_m
0.92	-0.39	-0.034	0.0045	-0.003
	-.67	-.051	.0059	-.008
	-.95	-.074	.0063	-.012
	-1.23	-.095	.0067	-.015
	-2.36	-.199	.0121	-.018
	-3.50	-.321	.0236	-.009
	.15	.004	.0050	.002
	.44	.027	.0054	.006
	.72	.048	.0059	.009
	1.00	.070	.0061	.013
	2.14	.174	.0114	.017
	3.28	.297	.0216	.008
	4.43	.434	.0400	-.020
	5.54	.553	.0642	-.052
1.20	-.47	-.028	.0143	.003
	-.74	-.050	.0146	.007
	-1.02	-.074	.0153	.010
	-1.28	-.093	.0160	.013
	-2.35	-.188	.0212	.026
	-3.41	-.281	.0301	.038
	-4.47	-.382	.0435	.053
	.05	.009	.0144	-.001
	.33	.032	.0142	-.004
	.58	.051	.0144	-.006
	.87	.074	.0150	-.009
	1.94	.164	.0196	-.019
	3.00	.252	.0274	-.031
	4.05	.336	.0386	-.042
	5.10	.418	.0526	-.052
	6.15	.500	.0692	-.062
	8.24	.654	.1099	-.082
1.50	-.27	-.018	.0157	.003
	-.53	-.031	.0158	.004
	-.78	-.047	.0160	.007
	-1.06	-.062	.0164	.009
	-2.12	-.126	.0195	.020
	-3.16	-.187	.0251	.030
	-4.20	-.248	.0331	.039
	.26	.011	.0153	-.002
	.53	.027	.0153	-.005
	.79	.041	.0157	-.006
	1.06	.057	.0163	-.008
	2.12	.122	.0199	-.020
	3.16	.185	.0259	-.030
	4.20	.246	.0342	-.040
	5.24	.305	.0447	-.051
	6.28	.364	.0570	-.061
	8.35	.479	.0880	-.080
	10.43	.594	.1279	-.102
	12.50	.693	.1718	-.117
1.50	14.58	0.802	0.2266	-0.136
	16.66	.902	.2879	-.151
	18.73	.995	.3557	-.164
1.70	-.26	-.014	.0157	.002
	-.52	-.026	.0153	.004
	-.79	-.040	.0155	.006
	-1.05	-.052	.0157	.008
	-2.10	-.104	.0183	.017
	-3.14	-.159	.0231	.027
	-4.18	-.211	.0298	.037
	.26	.011	.0149	-.001
	.52	.024	.0150	-.004
	.79	.037	.0151	-.005
	1.05	.050	.0152	-.008
	2.10	.102	.0177	-.017
	3.14	.156	.0226	-.027
	4.17	.209	.0291	-.037
	5.21	.260	.0376	-.046
	6.24	.311	.0480	-.055
	8.31	.413	.0747	-.074
	10.37	.513	.1080	-.093
	12.44	.614	.1497	-.113
	14.51	.700	.1944	-.127
	16.57	.791	.2484	-.144
	18.64	.881	.3098	-.160
	20.71	.965	.3771	-.174
1.90	-.26	-.014	.0161	.002
	-.52	-.024	.0153	.004
	-.78	-.037	.0152	.006
	-1.04	-.048	.0152	.007
	-2.09	-.094	.0177	.016
	-3.13	-.141	.0220	.024
	-4.16	-.189	.0281	.034
	.26	.007	.0140	-.002
	.52	.019	.0140	-.004
	.78	.030	.0142	-.006
	1.04	.042	.0145	-.007
	2.09	.088	.0171	-.016
	3.13	.136	.0215	-.025
	4.16	.183	.0278	-.034
	5.19	.228	.0352	-.042
	6.22	.274	.0444	-.051
	8.28	.367	.0685	-.069
	10.34	.456	.0987	-.087
	12.40	.549	.1363	-.106
	14.46	.632	.1791	-.121
	16.52	.716	.2281	-.137
	18.58	.797	.2835	-.153
	20.64	.874	.3447	-.169
	22.70	.944	.4105	-.183

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TABLE II.- AERODYNAMIC CHARACTERISTICS OF THE MODEL AT A REYNOLDS NUMBER OF 2.4 MILLION

M	α	C_L	C_D	C_m	M	α	C_L	C_D	C_m
0.60	-0.39	-0.028	0.0060	-0.004	0.75	0.44	0.024	0.0062	0.003
	-.67	-.045	.0064	-.006		.72	.043	.0064	.005
	-.95	-.065	.0069	-.007		1.01	.061	.0070	.007
	-1.22	-.081	.0076	-.009		2.14	.148	.0104	.012
	-2.34	-.157	.0108	-.012		3.26	.234	.0157	.016
	-3.43	-.230	.0167	-.016		4.39	.319	.0246	.021
	-4.54	-.308	.0262	-.023		5.51	.403	.0389	.026
	.15	0	.0058	-.001		6.63	.490	.0569	.027
	.43	.022	.0060	.002		8.80	.624	.0979	.003
	.71	.038	.0065	.004		10.87	.680	.1386	-.037
	.98	.055	.0071	.005					
	2.11	.134	.0104	.009	0.80	-.41	-.033	.0059	-.004
	3.20	.211	.0154	.012		-.69	-.052	.0066	-.006
	4.31	.286	.0240	.017		-.98	-.073	.0072	-.008
	5.40	.361	.0356	.021		-1.26	-.088	.0081	-.010
	6.50	.441	.0515	.020		-2.40	-.179	.0116	-.015
	8.69	.592	.0928	.007		-3.54	-.272	.0180	-.021
	10.73	.687	.1369	-.031		.15	.002	.0060	.001
	12.83	.717	.1753	-.060		.44	.025	.0060	.003
	14.84	.725	.2076	-.072		.73	.043	.0064	.006
	16.85	.729	.2388	-.076		1.01	.062	.0072	.008
						2.16	.150	.0104	.014
0.70	.15	0	.0060	.001		3.29	.245	.0164	.021
	.44	.022	.0060	.003		4.44	.339	.0265	.026
	.72	.040	.0059	.004		5.58	.438	.0429	.034
	1.00	.059	.0068	.006		6.74	.544	.0640	.037
	2.13	.140	.0098	.010					
	3.24	.221	.0147	.014	0.90	-.42	-.037	.0052	-.003
	4.35	.300	.0231	.019		-.71	-.055	.0058	-.008
	5.47	.385	.0367	.023		-1.00	-.077	.0063	-.011
	6.58	.467	.0536	.020		-1.28	-.094	.0071	-.015
	8.77	.612	.0947	.007		-2.45	-.199	.0113	-.023
	10.84	.677	.1363	-.035		-3.64	-.324	.0217	-.018
	12.88	.711	.1740	-.060		.15	.001	.0059	.004
	14.89	.715	.2053	-.067		.45	.027	.0063	.007
						.75	.048	.0066	.010
0.75	-.41	-.034	.0062	-.003		1.04	.068	.0073	.013
	-.69	-.053	.0065	-.005		2.20	.170	.0114	.022
	-.97	-.071	.0071	-.007		3.39	.294	.0211	.019
	-1.25	-.088	.0079	-.009		4.59	.428	.0388	.002
	-2.37	-.170	.0112	-.012		5.77	.558	.0633	-.027
	-3.51	-.258	.0171	-.017		6.90	.657	.0920	-.056
	.15	0	.0060	.001					

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TABLE II.- CONCLUDED

M	α	C_L	C_D	C_m
0.92	-0.43	-0.041	0.0056	-0.003
	-.72	-.058	.0060	-.007
	-1.01	-.083	.0063	-.010
	-1.30	-.104	.0070	-.012
	-2.46	-.211	.0123	-.016
	-3.66	-.341	.0245	-.004
	.15	0	.0061	.005
	.46	.029	.0062	.008
	.77	.053	.0065	.009
	1.06	.079	.0071	.011
	2.24	.191	.0130	.013
	3.43	.315	.0250	.001
	4.61	.441	.0441	-.023
	5.77	.550	.0663	-.045
1.20	-.48	-.031	.0144	.004
	-.78	-.053	.0148	.007
	-1.07	-.079	.0154	.011
	-1.34	-.099	.0162	.014
	-2.44	-.192	.0212	.027
	-3.53	-.284	.0303	.040
	-4.64	-.386	.0438	.056
	.06	.008	.0147	0
	.35	.034	.0148	-.003
	.63	.054	.0150	-.005
	.92	.077	.0155	-.008
	2.02	.171	.0201	-.021
	3.11	.260	.0278	-.033
	4.20	.344	.0394	-.044
	5.28	.426	.0536	-.053
	6.37	.509	.0713	-.063
1.50	-.28	-.018	.0153	.003
	-.56	-.034	.0154	.005
	-.83	-.051	.0152	.008
	-1.11	-.066	.0158	.010
	-2.18	-.129	.0197	.021
	-3.25	-.192	.0255	.032
	-4.32	-.255	.0337	.042
	.27	.009	.0157	-.001
	.54	.026	.0156	-.003
	.82	.041	.0160	-.006
	1.09	.058	.0166	-.008
	2.17	.123	.0200	-.019
	3.25	.187	.0261	-.030
	4.31	.248	.0343	-.040
	5.38	.309	.0448	-.051
	6.45	.371	.0578	-.062
	8.58	.482	.0890	-.080
1.50	10.70	0.594	0.1287	-0.098
	12.84	.705	.1773	-.118
	14.95	.802	.2306	-.136
1.70	-.27	-.014	.0135	.003
	-.54	-.027	.0138	.005
	-.82	-.042	.0141	.007
	-1.09	-.055	.0146	.009
	-2.16	-.108	.0175	.019
	-3.22	-.162	.0224	.029
	-4.28	-.217	.0292	.039
	.26	.008	.0140	-.001
	.54	.023	.0141	-.003
	.81	.036	.0144	-.005
	1.08	.050	.0149	-.008
	2.15	.104	.0181	-.018
	3.21	.159	.0233	-.028
	4.27	.213	.0304	-.038
	5.33	.264	.0393	-.047
	6.39	.319	.0505	-.058
	8.51	.420	.0781	-.076
	10.62	.521	.1135	-.094
	12.75	.622	.1567	-.112
	14.87	.719	.2065	-.131
	16.23	.776	.2523	-.143
1.90	-.27	-.016	.0146	.003
	-.54	-.028	.0144	.005
	-.81	-.040	.0145	.007
	-1.08	-.049	.0148	.008
	-2.14	-.098	.0173	.017
	-3.19	-.144	.0215	.026
	-4.24	-.191	.0273	.035
	.26	.004	.0134	-.001
	.53	.017	.0135	-.003
	.80	.029	.0137	-.005
	1.06	.040	.0140	-.007
	2.13	.088	.0167	-.016
	3.18	.136	.0210	-.025
	4.23	.182	.0272	-.033
	5.28	.227	.0351	-.042
	6.33	.274	.0448	-.051
	8.43	.367	.0693	-.070
	10.53	.455	.1001	-.086
	12.64	.544	.1381	-.103
	14.75	.634	.1838	-.119
	16.86	.721	.2358	-.134
	18.44	.787	.2800	-.148

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TABLE III.- AERODYNAMIC CHARACTERISTICS OF THE MODEL AT A REYNOLDS NUMBER OF 3.8 MILLION

M	α	C_L	C_D	C_m
0.60	-0.42	-0.033	0.0071	-0.003
	-.71	-.051	.0073	-.004
	-1.00	-.069	.0077	-.005
	-1.28	-.085	.0081	-.007
	-2.40	-.160	.0108	-.011
	-3.53	-.239	.0169	-.016
	-4.68	-.321	.0271	-.022
	.15	.005	.0071	0
	.44	.021	.0072	.003
	.73	.039	.0072	.004
	1.02	.058	.0077	.006
	2.16	.134	.0093	.010
	3.28	.210	.0136	.014
	4.42	.290	.0222	.019
	5.54	.365	.0352	.024
	6.68	.442	.0511	.027
	8.92	.596	.0928	.013
	11.07	.700	.1398	-.028
0.70	-.43	-.034	.0066	-.003
	-.72	-.053	.0070	-.005
	-1.02	-.074	.0074	-.006
	-1.30	-.091	.0080	-.008
	-2.45	-.171	.0111	-.013
	-3.60	-.253	.0175	-.018
	.15	.001	.0068	.001
	.45	.024	.0067	.003
	.75	.042	.0067	.005
	1.05	.063	.0071	.007
	2.20	.144	.0091	.012
	3.35	.228	.0141	.017
	4.51	.309	.0239	.022
	5.65	.388	.0377	.029
	6.79	.471	.0552	.028
	9.06	.625	.0987	.011
	11.19	.710	.1441	-.036
	13.22	.736	.1822	-.066
0.75	-.44	-.037	.0065	-.003
	-.73	-.054	.0070	-.005
	-1.03	-.074	.0077	-.007

M	α	C_L	C_D	C_m
0.75	-1.31	-0.093	0.0081	-0.008
	-2.47	-.177	.0114	-.013
	-3.64	-.263	.0160	-.021
	.15	.002	.0068	.001
	.46	.027	.0068	.003
	.76	.045	.0068	.005
	1.06	.067	.0074	.007
	2.22	.151	.0094	.013
	3.38	.235	.0145	.019
	4.56	.324	.0253	.027
	5.72	.409	.0394	.034
	6.88	.495	.0575	.035
	9.13	.644	.1030	.006
0.80	-.44	-.037	.0062	-.003
	-.75	-.057	.0064	-.005
	-1.04	-.079	.0074	-.007
	-1.33	-.098	.0078	-.009
	-2.50	-.185	.0110	-.016
	-3.70	-.283	.0190	-.026
	.15	.001	.0065	.001
	.47	.027	.0066	.004
	.77	.046	.0068	.006
	1.08	.069	.0072	.008
	2.26	.161	.0098	.015
	3.44	.251	.0156	.024
	4.64	.349	.0279	.034
	5.83	.451	.0445	.041
	7.05	.564	.0668	.045
0.90	-.46	-.041	.0058	-.004
	-.77	-.065	.0061	-.007
	-1.07	-.088	.0069	-.010
	-1.37	-.111	.0076	-.012
	-2.59	-.214	.0125	-.022
	-3.86	-.345	.0253	-.018
	.15	.006	.0063	.003
	.49	.033	.0058	.005
	.80	.053	.0062	.009
	1.11	.081	.0066	.012
	2.34	.192	.0108	.019
	3.61	.320	.0212	.018

TABLE III.- CONCLUDED

M	α	C_L	C_D	C_m
0.92	-.47	-.045	.0058	-.003
	-.78	-.067	.0062	-.006
	-1.08	-.092	.0068	-.009
	-1.39	-.119	.0078	-.011
	-2.64	-.242	.0146	-.010
	-3.91	-.376	.0293	.003
	.17	.007	.0063	.003
	.50	.034	.0058	.006
	.82	.060	.0062	.009
	1.12	.086	.0065	.011
	2.37	.209	.0118	.012
	3.85	.348	.0260	-.007
	4.87	.460	.0450	-.025
1.20	-.54	-.037	.0152	.006
	-.84	-.060	.0157	.009
	-1.13	-.086	.0163	.013
	-1.41	-.109	.0170	.016
	-2.56	-.200	.0226	.029
	-3.71	-.293	.0327	.041
	.07	.009	.0145	0
	.39	.037	.0145	-.003
	.65	.062	.0147	-.006
	.98	.087	.0154	-.010
	2.13	.180	.0201	-.022
	3.28	.269	.0285	-.033
	4.42	.354	.0403	-.041
	5.57	.439	.0554	-.052
	7.11	.556	.0795	-.066
1.50	-.30	-.021	.0154	.004
	-.60	-.038	.0157	.007
	-.88	-.056	.0161	.010
1.50	-1.16	-0.072	0.0166	0.012
	-2.27	-.136	.0205	.023
	-3.38	-.199	.0269	.034
	-4.49	-.262	.0361	.044
	.28	.010	.0153	0
	.57	.028	.0154	-.003
	.86	.045	.0157	-.006
	1.15	.062	.0163	-.009
	2.26	.131	.0200	-.021
	3.37	.194	.0262	-.031
	4.47	.253	.0347	-.041
	5.58	.315	.0456	-.051
	6.69	.376	.0589	-.059
	8.90	.496	.0927	-.080
1.70	-.29	-.017	.0141	.003
	-.57	-.030	.0144	.005
	-.86	-.045	.0148	.008
	-1.14	-.059	.0152	.011
	-2.23	-.115	.0184	.021
	-3.32	-.170	.0237	.031
	-4.40	-.223	.0310	.041
	.27	.010	.0141	-.001
	.56	.025	.0143	-.004
	.84	.039	.0145	-.006
	1.13	.054	.0150	-.009
	2.22	.112	.0182	-.020
	3.30	.166	.0234	-.030
	4.39	.219	.0305	-.040
	5.48	.271	.0397	-.049
	6.56	.324	.0511	-.058
	8.73	.426	.0795	-.075
	10.91	.533	.1170	-.095

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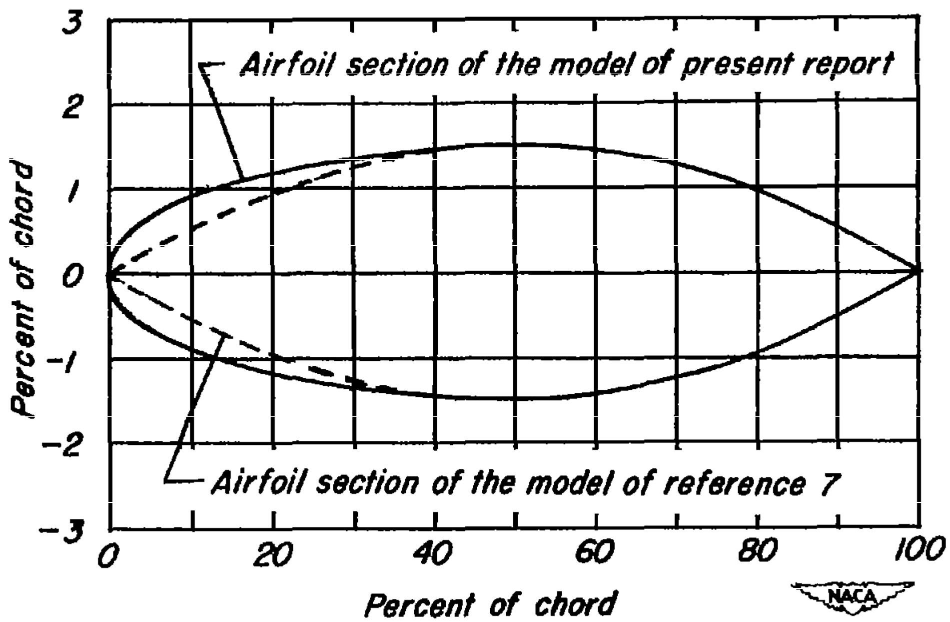


Figure 1.- Comparison of the airfoil section of the model of present report with that of reference 7.

[redacted]

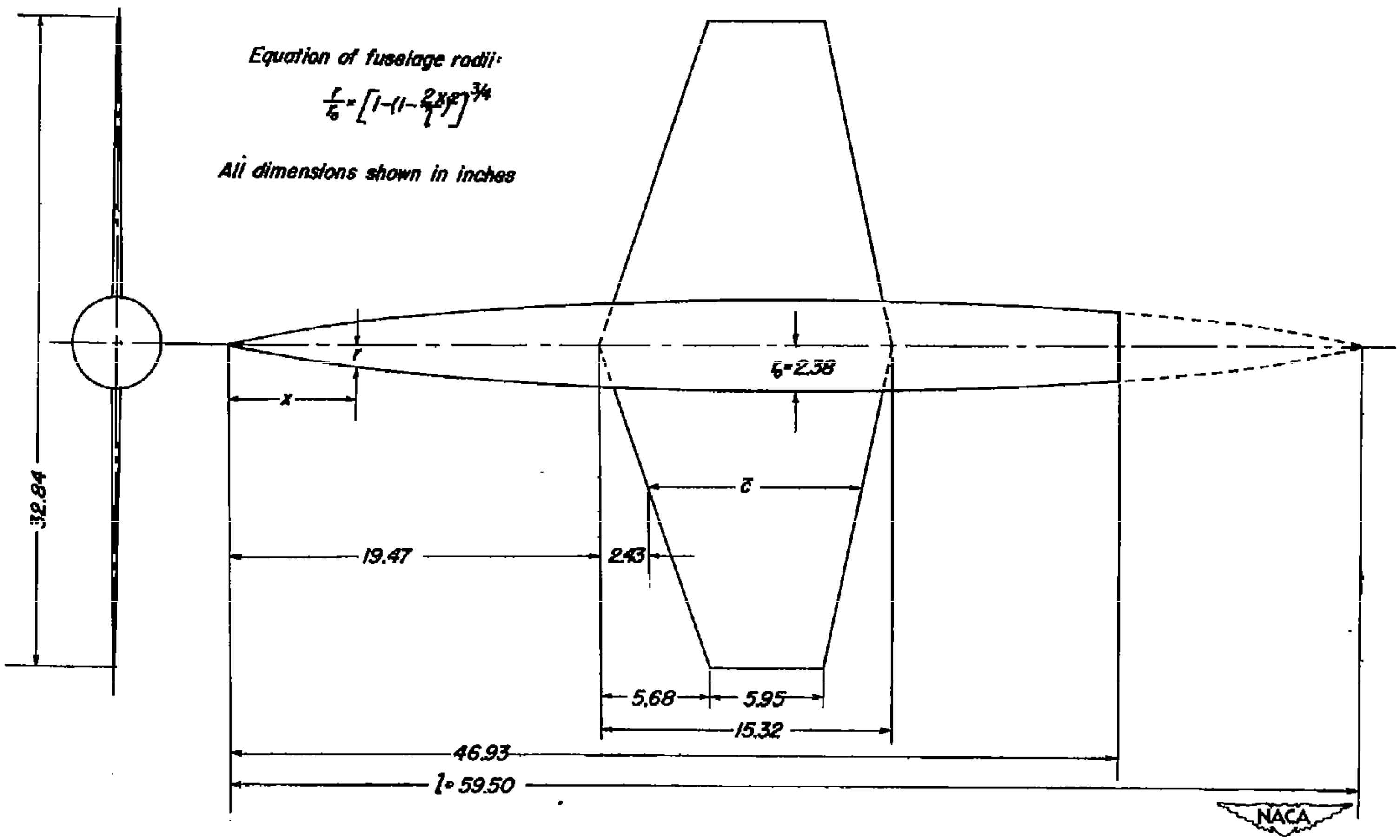
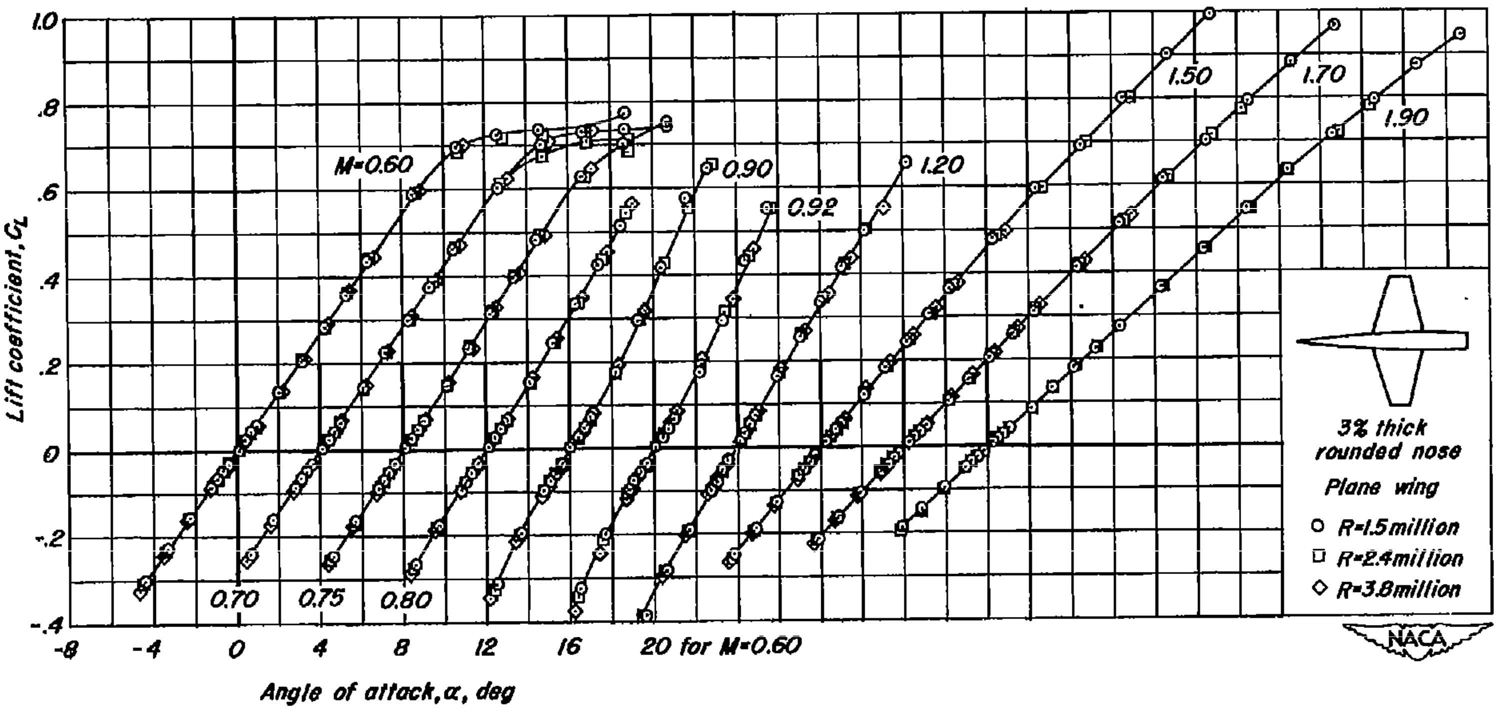
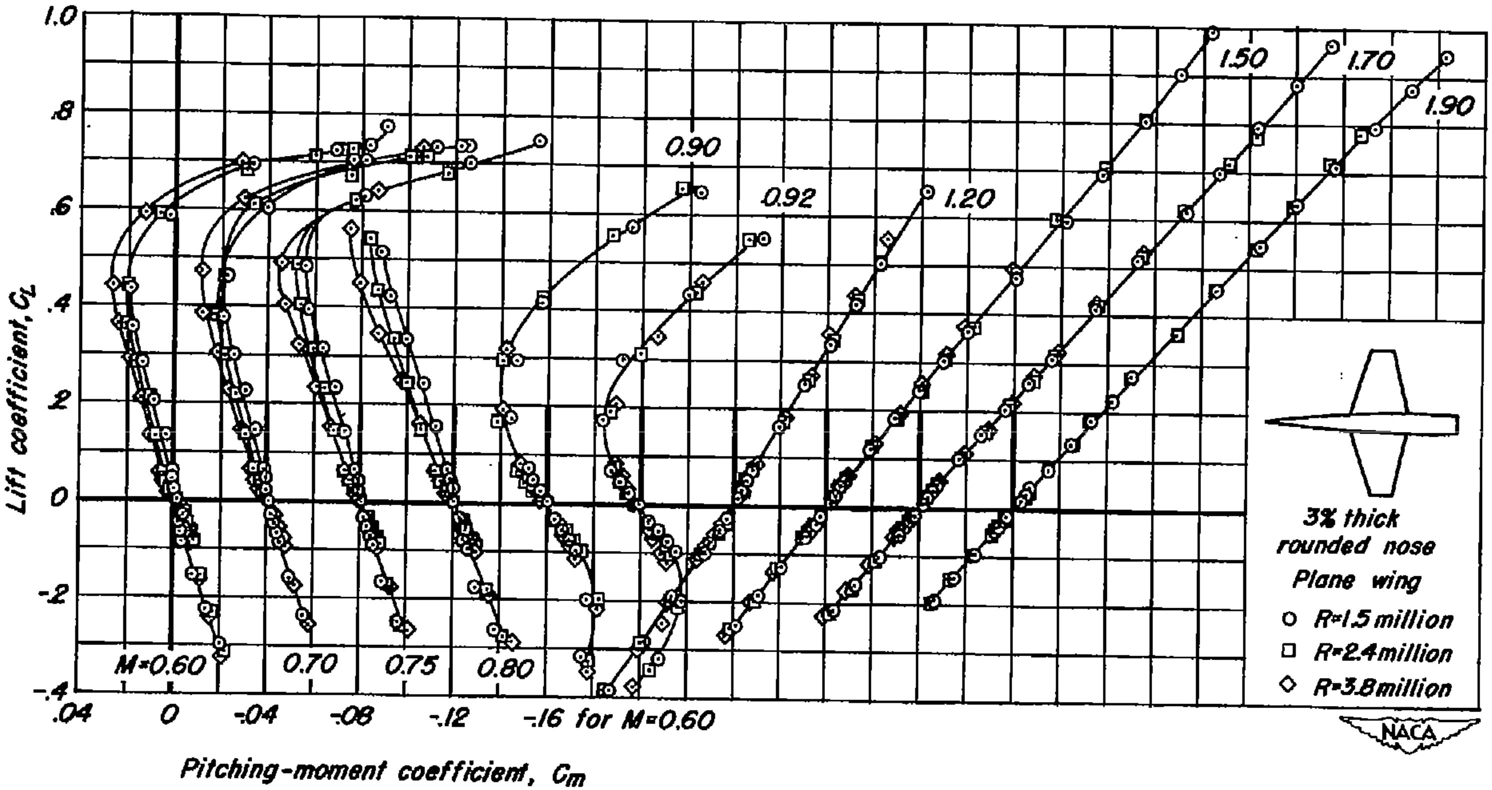


Figure 2.– Plan and front views of the model.



(a) C_L vs α

Figure 3.–The variation of the aerodynamic characteristics with lift coefficient at various Mach numbers.



(b) C_L vs C_m

Figure 3.-Continued.

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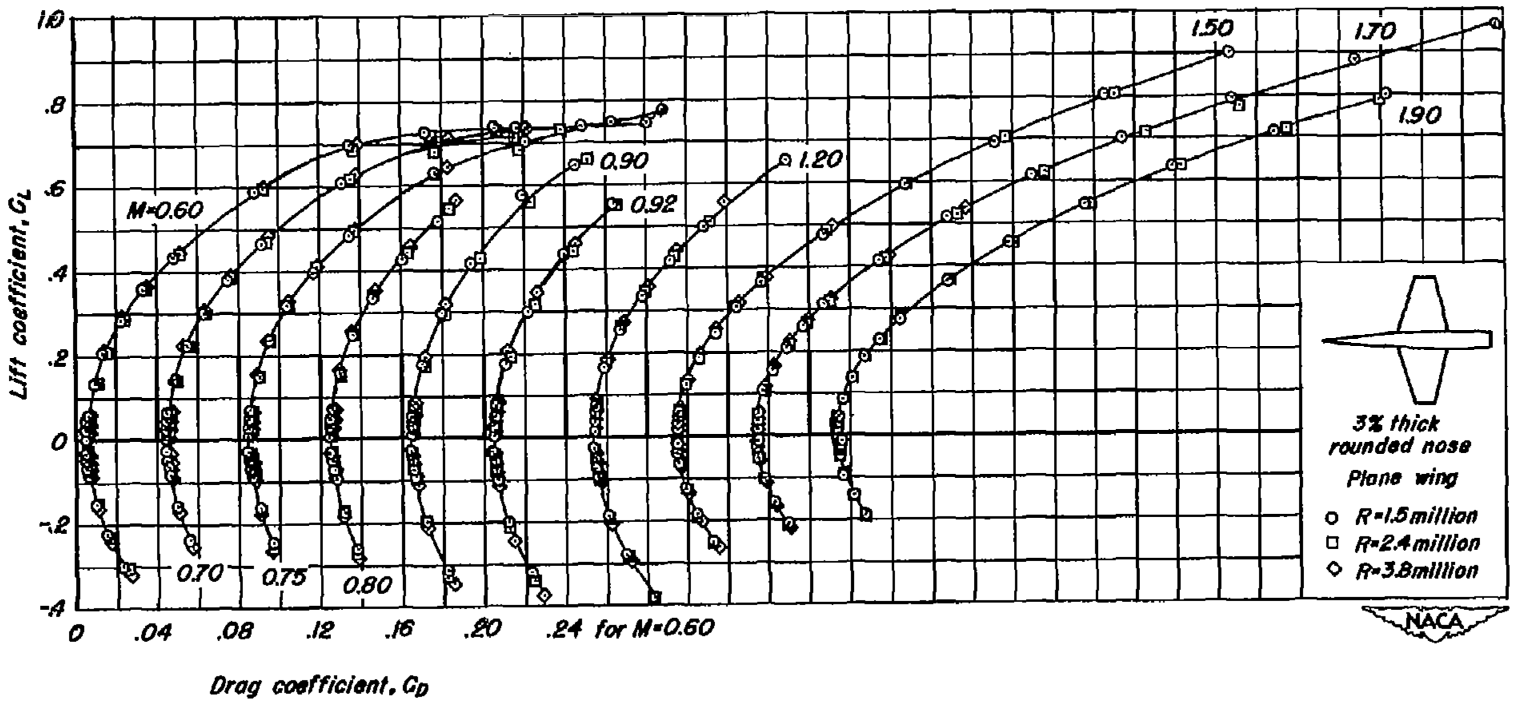
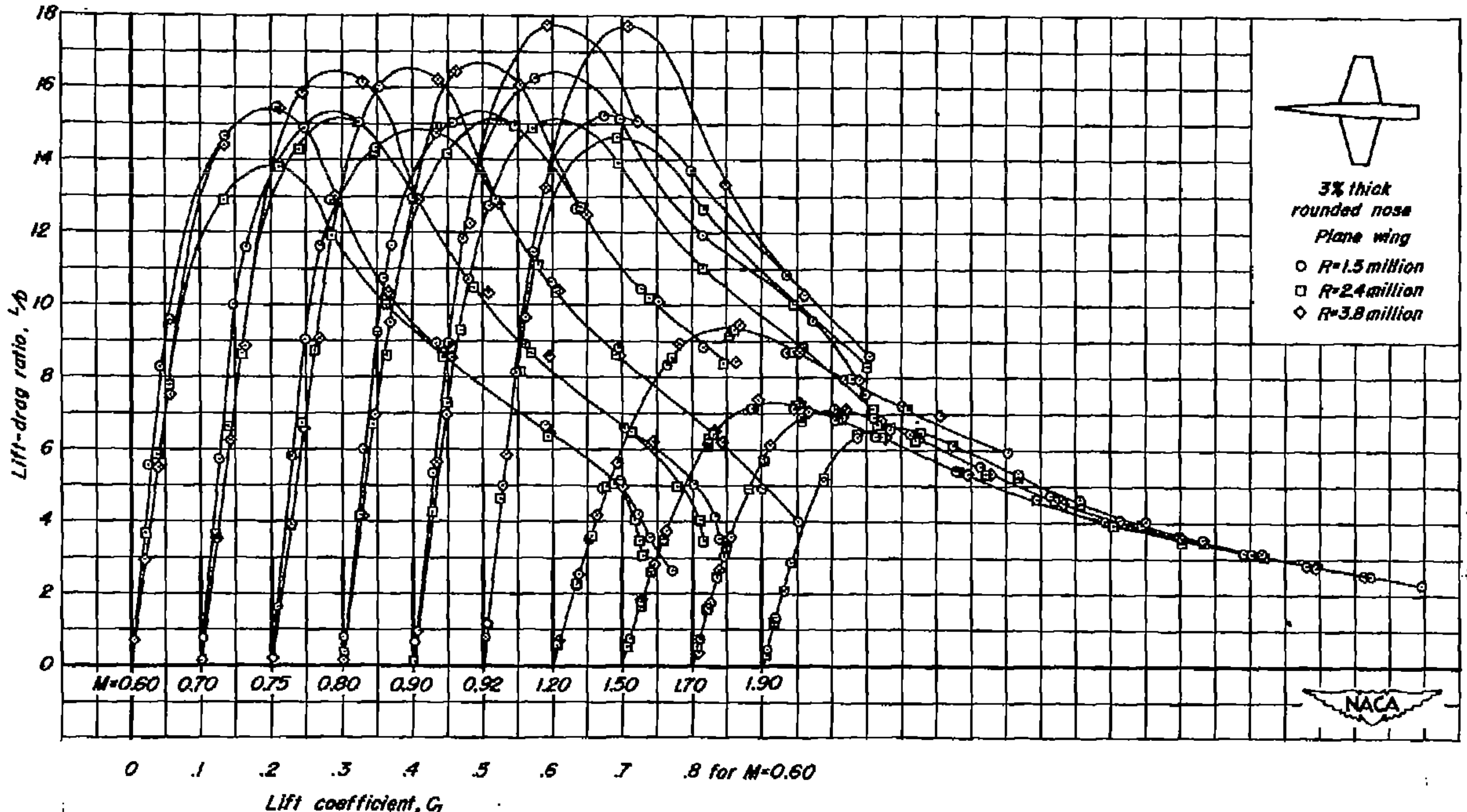


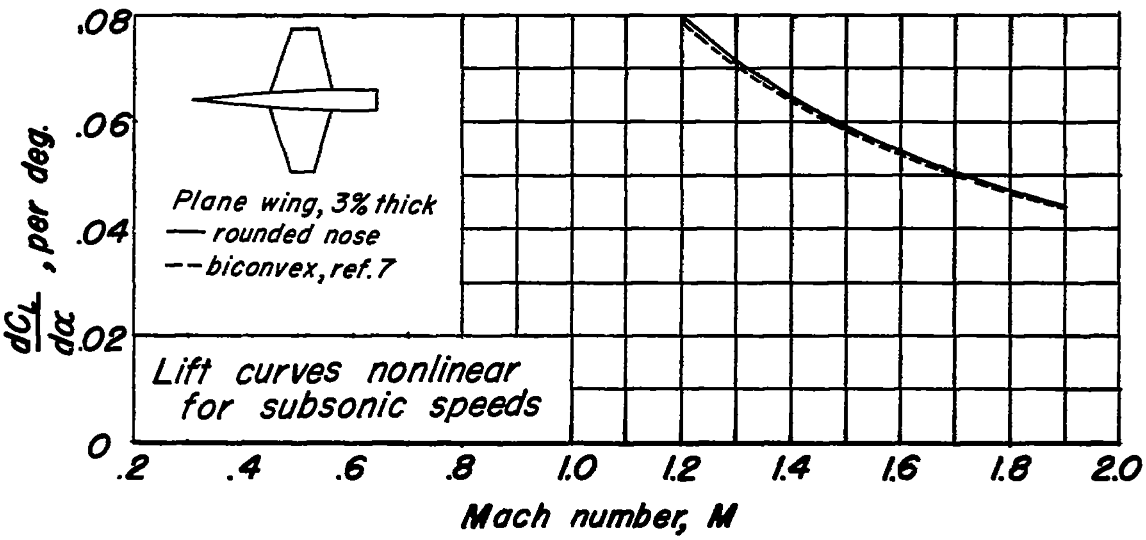
(c) C_L vs C_D

Figure 3.- Continued.

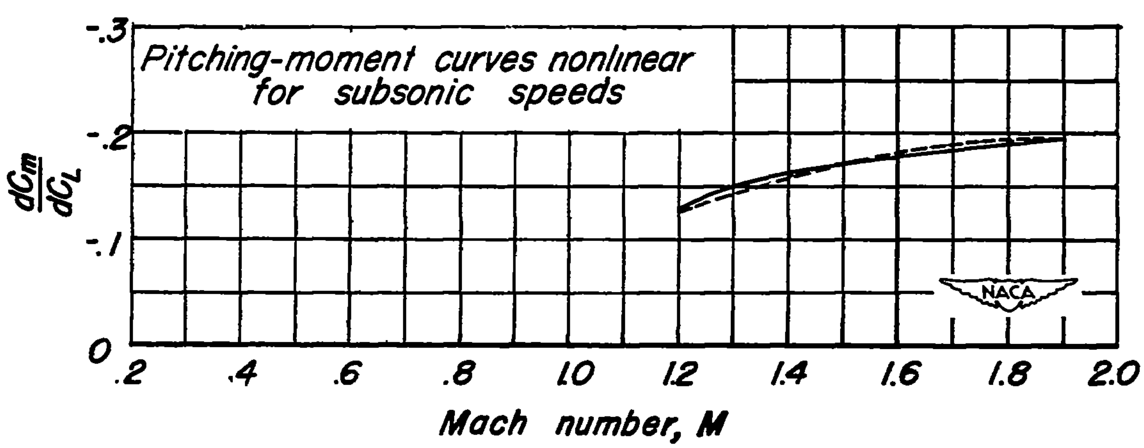


(d) L/D vs C_L

Figure 3.—Concluded.

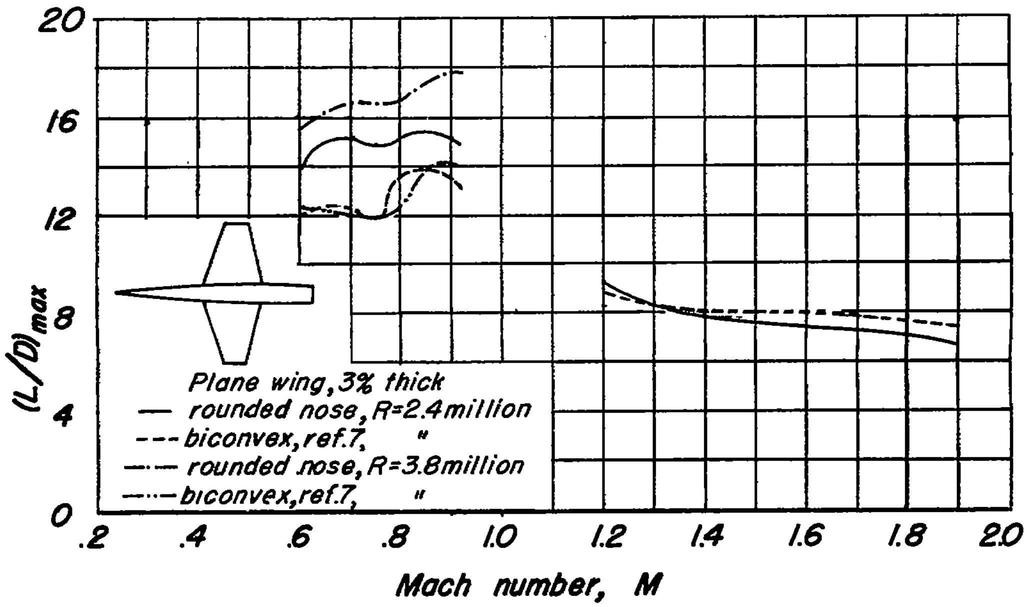


(a) $\frac{dC_L}{d\alpha}$ vs M

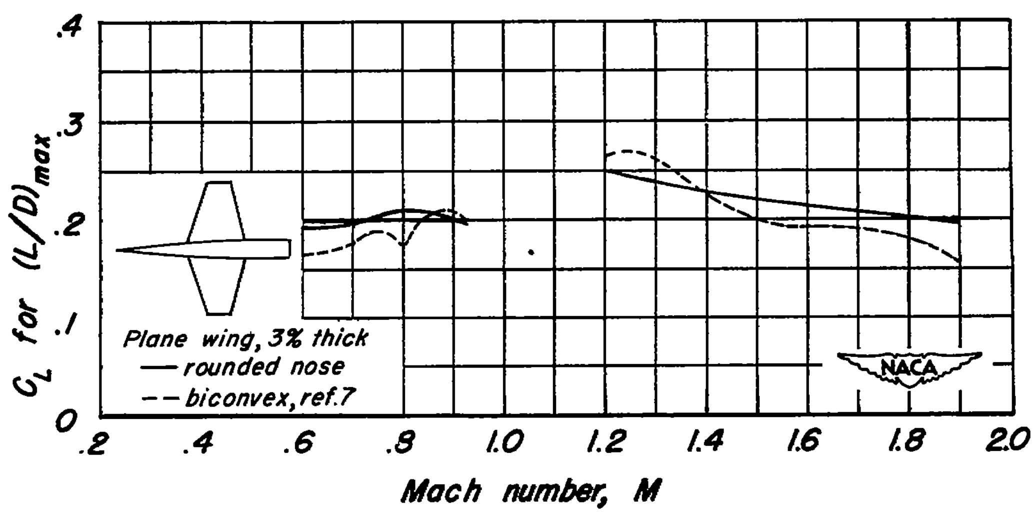


(b) $\frac{dC_m}{dC_L}$ vs M

Figure 4.- Summary of aerodynamic characteristics as a function of Mach number. Reynolds number, 2.4 million.

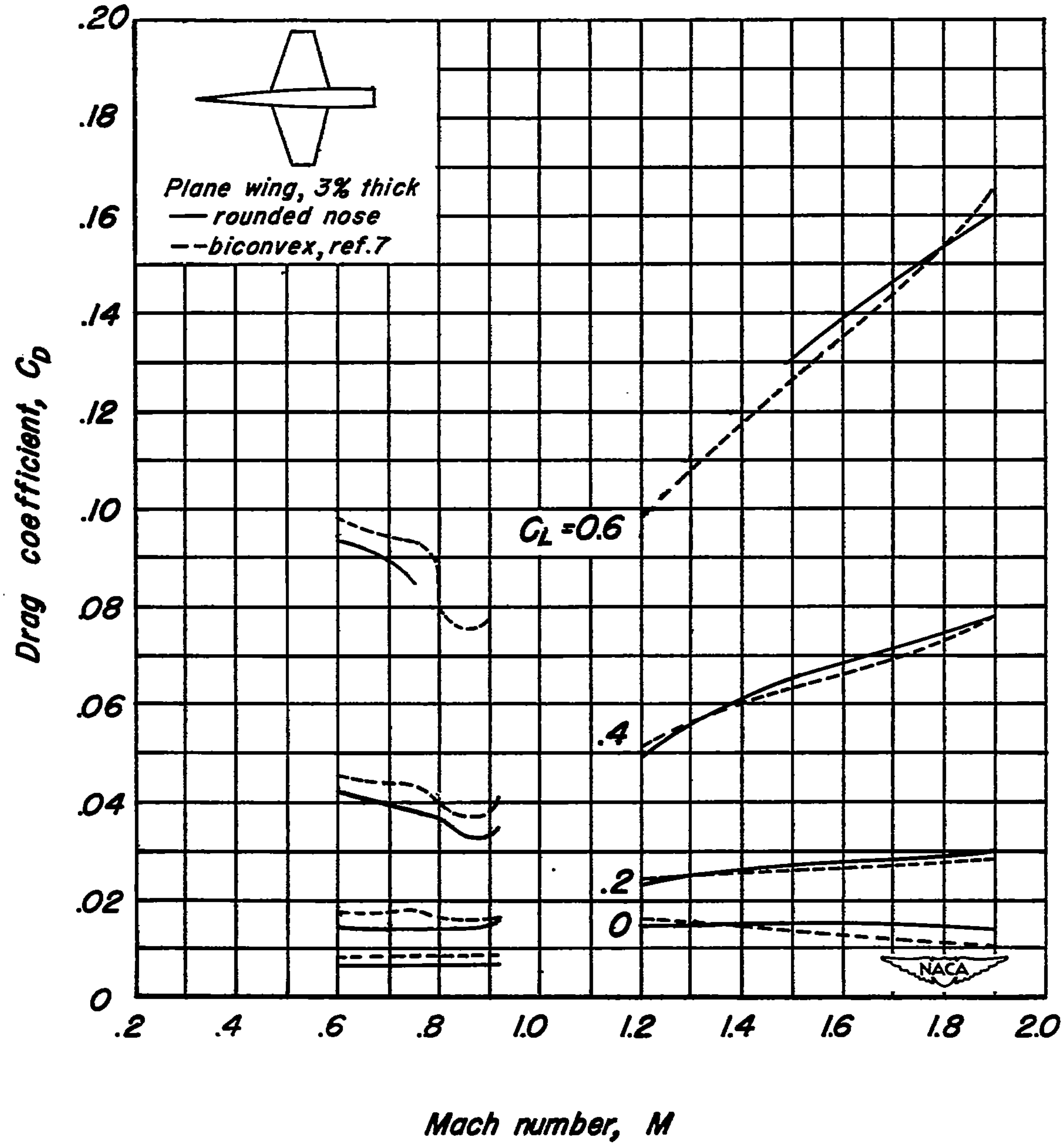


(c) $(L/D)_{max}$ vs M



(d) C_L for $(L/D)_{max}$ vs M

Figure 4.-Continued.



(e) C_D vs M

Figure 4.-Concluded.

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